TEMPERATURE CONTROL OF UNMANNED SPACECRAFT

by

Milton Schach and Robert E. Kidwell, Jr.
NASA Goddard Space Flight Center
Greenbelt, Maryland

42—SPACE

Bond (11) 9 on 11 Optima x 18 — Space — JULY
Art. No. 59 — Heat Balance — LD/fk — 6-14-65

Temperature control of spacecraft depends on well-known principles of heat transfer, and it might appear that space vehicle thermal design should be a completely straightforward process. But the variety and complexity of spacecraft, the difficulties in accurate experimental verification, and uncertainties in energy transfer between surfaces—all these along with the effects of the space environment on surface properties—call for imaginative design judgment and advances in many supporting technologies.

The first generation of American earth satellites were small, highly symmetrical in shape, simple in design, and planned for missions of short duration. Temperature control was based primarily on the use of surface coatings and finishes that would provide the required balance between energy received (mainly from sunlight) and energy reradiated to space in about the same way that the earth's temperature is maintained.

The feasibility of spacecraft temperature control could be seen in the fact that the surface temperatures of an old satellite of the sun—the earth—remain within the fairly narrow range of about −40 to 40 C, with the mean close to 14 C. The earth absorbs an average of nearly two-thirds of the incident sunlight and radiates the same amount of energy. Although the earth's atmosphere reduces the rate at which heat is returned to space, the earth's surface and its atmosphere, together, are in thermal equilibrium with the sun's radiation.

The heat balance is somewhat different for an artificial satellite, of course. The earth is nearly spherical and spins on an axis which is 23½ deg from being perpendicular to the orbital plane; moreover, the energy from the warm interior of the earth does not contribute significantly to the temperature of its surface. An artificial satellite may or may not be spherical and it may or may not spin. If it does spin, the orientation of the spin axis may not be fixed relative to the ecliptic plane. The temperature of its surfaces may be significantly affected by internal heating. And a satellite in orbit about the earth may be totally eclipsed for periods long enough, and with sufficient frequency, to cause substantial changes in temperatures even at interior points. In addition, the satellite can receive energy from the earth's surface in the form of emitted radiation and reflected sunlight. If a spacecraft probes toward the planets, the solar intensity will vary as the inverse square of distance. And finally, the satellite does not carry with it its own atmosphere to reduce heat loss to space and to provide protection from energetic particles, ultraviolet radiation, micrometeoroites, or other effects of the space environment which are harmful to materials.

The technology for temperature control of early spacecraft was primitive. Analytical tools were undeveloped and little was known about the magnitude of space effects or what would happen to surface materials used for temperature control. Test equipment for simulating the solar-space environment to verify a thermal design was virtually nonexistent. Fortunately, our knowledge of radiation from the sun and the earth and our instrumentation for determining the absorptivity and emissivity of materials were sufficiently accurate for thermal design requirements at that time.

Vanguards 1 and 2, for example, were spherical shells surrounding a small internal canister which housed the electronics. While a temperature gradient could exist on the surface of the spherical shell (for about the same reason that temperature varies with latitude over the earth's surface), it was expected that the inner package, isolated from the shell, would attain a temperature close to the mean temperature of the shell. This mean temperature would depend on the average energy absorbed from the sun and earth during each orbit. And this energy would change only as the position of the orbit relative to the sun and earth varied slowly day by day. Because of the spherical shape, the absorbed energy would not change with the orientation of the spin axis.

43—SPACE AERO

WATERS

Art. No. 39 — Heat Balance — LD/ik — 6-14-65

The early Explorers, on the other hand, were cylindrical and it was anticipated that the absorbed energy would vary with both the attitude of the spin axis and position of the orbital plane. The possible variations of temperature could have approached or exceeded the temperature limits of critical components. One of the conclusions drawn from Explorer 1 temperature data was that the use of coatings for passive temperature control was limited to applications where there wasn't too much total variation in absorbed solar energy. The solution was to restrict the time of day of launch so that the initial orientation and subsequent changes would be known and thermal coating patterns could be determined accordingly.

Spacecraft have since grown in size, weight and complexity, and the diversity of current space missions has given rise to a variety of thermal control systems. These have in common the aim of modifying the heat transfer to and from each spacecraft element so that its allowable temperature range will not be exceeded during the entire life of the mission. To achieve this the key initial design technique is to create a thermal model of the spacecraft. This model is a mathematical representation of the process of satellite thermal energy balance.

The first step is to subdivide the spacecraft into small isothermal elements, then accurately describe the heat balance for each of the elements. The designer's experience and judgment are used to select the elements needed to obtain the accuracy required and to minimize engineering and computer time. To assist in detailed thermal analysis of spacecraft designs, extensive use is made of large, high speed digital computers such as the IBM 7094. A variety of basic programs have been written to perform many calculations that would be impractical or impossible to do by hand.

There are, typically, two modes of energy transfer—conduction and radiation. The heat transferred from an element to an adjacent element through solid material is simply a function of dimension and thermal conductivity, but conduction across a joint in a vacuum environment is a poorly understood process. At the present time, when quantitative data are essential for thermal design, it is necessary to derive these data from tests on actual hardware.

The radiation transfer is of two classes. First, there is radiation from external sources such as the sun, earth, other planets and the moon. Second, there are the radiation exchanges between the elements of the satellite itself. These radiation sources contribute in distinct ways to the energy balance because of their different intensities and difference in spectral content. The spectral distribution of energy is important because materials absorb radiation differently in different wavelength regions.

The sun has an intensity of 130 watts per sq ft (perpendicular to the direction of propagation) at the earth's mean distance from the sun. Approximately 98 percent of this intensity lies in the wavelength region between 0.3 and 4.0 microns.

The radiation from the earth is generally typical of the planets. It consists of earth-reflected solar radiation and earth-emitted radiation. The eatrh reflects or scatters, on the average, about one-third of incident solar radiation. In wavelength distribution, this reflected radiation is roughly similar to direct solar radiation. Its intensity incident on a satellite element depends on satellite position with respect to the sunlit half of the earth. At a height of 20,000 miles above the earth, albedo radiation reaching the satellite falls to about one percent of direct solar energy. For low orbits and satellite positions over the center of the sunlit hemisphere, the intensity reaches 50 percent.

The earth's surface and atmosphere absorb the other two-thirds of incident sunlight, and because the earth is in equilibrium it must re-emit this absorbed radiation. The secondary radiation, emitted at an average temperature near —23C, lies in the wavelength region approximately between 5 and 50 microns. The intensity of this radiation also depends on the height of the satellite above the earth and its orientation.

Radiation emitted by elements of the satellite itself is important not only because it is an additional energy source but also because, from the point of view of an emitting element, it constitutes an energy loss. The energy emitted is proportional to the fourth power of the temperature. It depends, in addition, on a characteristic surface parameter, thermal emittance, which varies widely for various satellite thermocoating materials.

The effect of the various types of radiation on thermal balance of a satellite element depends on the solar absorptance for the particular thermocoating material—that is, the fraction of incident solar radiation absorbed by a surface which varies with different materials. The absorptance of a surface for radiation emitted by the earth and for that emitted by the elements of the satellite are each taken to be numerically equal to the thermal emittance of the surface. This approximation is adequate for most surfaces.

WATERS
Art. No. 59 — Heat Balance — LD/fk — 6-14-65

In carrying out practical thermal control designs for spacecraft, we use passive or active techniques, or both. The thermal design is largely dependent on the spacecraft's mission. Passive control, using coatings of various types, has been used primarily for small spacecraft either in near-earth orbits or for probes which remain approximately 1 AU (the mean distance between the earth and sun) from the sun. Passive temperature control is inherently reliable since no moving parts are required, and it adds the least additional weight. These two advantages are easily understood by project managers because minimum weight and high reliability are among the most desired features of a spacecraft design.

The disadvantages are apparent to thermal designers but are less well understood by others. The basic limitation of passive temperature control is that once the spacecraft is in orbit the temperature is determined solely by the energy balance. Typically, energy balance changes with time and a corresponding range of temperatures is experienced. Passive design attempts to place this range in an allowable region. The width of the average temperature range will depend on the extent of change in internal heat dissipation and in a number of orbital characteristics, including: the fraction of the orbital period in sunlight; effective cross section area of the satellite for solar radiation; incident planetary radiation; and distance between the earth and sun. For a spherical satellite in a near-earth polar orbit, the average temperature would lie within a band about 30 C in width. For a cube in a similar orbit, the effect of the changing cross-section for solar radiation would increase the temperature bandwidth to 60 C.

Thus far we have talked about average temperatures. In addition, there would be temperature gradients on external surfaces and to a lesser extent among interior parts. Therefore, total temperature changes and uncertainties would be greater than those cited above. There are other uncertainties, too. Measurements of solar absorptance and emittance of thermal coatings may be in error, quality control in application of the coatings may vary, and coatings may be degraded by preflight handling, during launch and by the space environment. Furthermore, there are likely to be errors and approximations in the design analysis. What happens in practice is that the designer is often not conservative enough in his estimates; he may underestimate the error tolerances, widen the temperature limits, and not sufficiently restrict the range of expected orbits or attitudes. As a result, he is faced with predicted temperature variations which approach or exceed specified temperature limits.

One way to reduce uncertainties and narrow the temperature ranges electronics components must face—and thus improve reliability—is to use active rather than passive temperature control. An example of this philosophy is the thermal design of the Telstar satellite. The almost spherical structure, thermal isolation of critical components in an internal canister, and a near-earth orbit made it quite similar to Vanguards 1 and 2. A temperature-actuated lid on an insulated canister housing the electronics was employed to maintain normal room temperature for maximum reliability. Another example is the thermal design for Nimbus-1. Although this highly nonspherical, complex structure was launched in a near-earth orbit, the fact that it was earth-oriented and that it was always to be in a noon orbit made it conceivable that passive temperature control would have worked. However, active control using a system of external louvers was specified as a design objective for maximum reliability.

Active control is mandatory for some missions. At present three basic methods are used for control of spacecraft temperature: variable internal conduction or radiation transfer to the outer shell; adjustable external surfaces; and electric heaters. For the Mariner spacecraft, the variation in solar intensity between earth and Venus or between earth and Mars would be too large for passive methods. At Jupiter and other planets farther from the sun, solar intensity is negligible for temperature control use. Another source of energy, such as a radioisotope, is required to provide heat for temperature control. For trips to Mercury, or nearer the sun, sunshades may be required to reduce the effective solar intensity on the spacecraft.

Another mission requiring active temperature control is that of the Orbiting Geophysical Observatories, such as OGO-1, which may be used in a variety of orbits, ranging from nearly circular orbits a few hundred miles above the earth to highly elliptical orbits (approximately 200 miles perigee and 70,000 miles apogee). The basic spacecraft design structure is essentially a "boxcar" that will admit a wide variety of experiment packages to be mounted in the spacecraft, or on booms and star paddles. Since the satellite is intended to serve as an instrumentation platform for experiments that cannot be anticipated during space vehicle design, active control of the experiment packages is mandatory.

Active temperature control may also be required fofr a component part of a subsystem with unusual temperature limits. For example, to maintain the stable frequency of an oscillator, temperature control to within a fraction of a degree may be necessary. Similar control of temperature gradients in a telescope may be required to prevent misalignment of the optics. Other components may have temperature limits substantially different from the usual limits for batteries and electronic components. The diverse and narrow-range temperature requirements for two components of the rubidium-vapor magnetometer aboard the IMP-1 satellite were typical. The magnetometer's gas cell required temperature limits of 37 to 47 C, and its lamp required 100 to 120 C, while the spacecraft's fluxgate magnetometer could tolerate a range of —50 to 50 C and the electronic cards 0 to 35 C.

Since electric power is rationed on a priority basis, the use of electric heaters is generally limited to small components. Passive control is used to insure that without heater power the nominal temperature desired would not be exceeded under any possible orbital condition. Heaters are turned on when temperature drops below a specified level and turned off at a higher level. Simple on-off controls such as thermostats are generally preferred. Insulations, such as lightweight multilayer aluminized Mylar blankets, are used to minimize power requirements.

Temperature control by adjustable louvers is essentially a matter, used by OGO-1 and Nimbus 1, of changing external surface properties. (Pegasus uses louvers for regulating heat transfer from the spacecraft to the cool rocket casing which remains attached to the craft.) One use of louvers is to locate them on surfaces that are always shaded from direct sunlight. The louvers may be insulated or coated with a low emittance surface, such as evaporated aluminum, so that in a closed position their heat loss to space is minimized. Behind the louvers is a surface of high emittance, usually white paint. As temperature of the spacecraft rises, the louvers are made to open gradually and increase the heat losses to space by allowing some of the radiation from the white plate to escape. Temperatures may be sensed by bimetallic springs directly mounted to the louvers or by various other methods. The effect of the louvers can be maximized by insulating other surfaces of the spacecraft.

There are problems with louver systems, of course —particularly those connected with vibration levels during launch, calibration, bearings, added weight to the spacecraft, and lack of data on the effect of incident sunlight on louver surfaces.

The technique of active control using variable internal conduction is analogous to the use of electric heaters except that the waste heat of electronics is employed instead, along with required thermomechanical techniques. Thermal coatings are used to passively maintain shell temperatures below the nominal operating temperature range for the internal compartment. The interior compartment is carefully insulated from the shell to minimize heat transfer from the electronics to the shell except by controlled conduction and radiation paths. Thus, a gradient is established to maintain the interior temperature above the shell temperature.

To calibrate the controller and to verify the overall thermal design for this, as well as other methods, tests in a simulated space environment are necessary. There are two general approaches to simulation. One method, which simulates solar (and planetary) energy absorbed by the satellite surface, uses electric heater windings mounted on or just below the outer surface. For spacecraft with simple geometric shapes, this technique has considerable merit and convenience. The other method, a more direct and promising technique known as solar simulation, employs a light source and optics to product a uniform collimated beam of an intensity equal to one solar constant. The desired results call for a spectral match, in addition to good collimation and uniformity.

Thermal control of spacecraft has worked surprisingly well in most cases, but in some instances severe restrictions have been placed on the mission in order to stay within desired temperature limits. More precise spacecraft temperature control is essential for higher reliability and longer-life missions. To achieve it, we will have to show progress in overcoming limitations in analytical and experimental techniques and difficulties in protecting the satellite from hostile environments during preflight tests, launch and spaceflight.

The complexity of satellite structure makes it necessary for the designer to use a simplified model for his analysis. The subdivision of an elaborate structure into a few dozen, or at most a few hundred, isothermal elements, inevitably introduces uncertainty that is frequently difficult to evaluate. A similar uncertainty arises in the computation of radiation exchange between external or internal elements because of the complicated geometric arrangement and the imperfectly diffuse reflectance of the surfaces.

More uncertainty is contributed by the unpredictable transfer of heat across interfaces. The numerous methods and materials used in mounting, supporting and connecting equipment produce diverse interface conditions. Poor reproducibility of interface conditions is a persistent problem.

Perhaps the greatest uncertainty is in our knowledge of surface properties—solar absorptance and thermal emittance—of thermocoating materials. The uncertainty is due to limitations in measurement technique, initial differences in nominally identical surface coatings, and subsequent changes in materials before, during and after launch.

The variability in surface coatings depends on the type of polished metal, evaporated metal and oxide, or paint. Considerable difficulty has been experienced with marginal reproducibility in paints and polished metals. Other coatings, in addition to handling damage, are subject to possible contamination in vacuum systems during tests or from release of gas by the vehicle fairing during launch. Further changes in coating may occur in orbit due to the sun's ultraviolet radiation and to charged particles, which degrade the optical properties of most materials.

48—SPACE AERO

KABEL—JUNE 14—9-11 Optima x18
Art. No. 59 — Heat Balance — LD/fk — 6-14-65

Future improvements may be expected to occur in all aspects of spacecraft thermal design. Analysis will become more accurate with the availability of computers having larger storage capacity and, thus able to handle more detailed studies. In passive thermal control, organic and inorganic coatings with greater stability under ultraviolet radiation will be developed, and so will better procedures for protecting coatings from ground and launch contamination. The effects of charged particles on coatings will be better understood.

Refinement of laboratory techniques for the measurement of solar absorptance and thermal emittance will continue. Careful measurements of these properties in space will provide direct evidence of the accuracy of measurements and of their changes in the orbital environment. Studies of the process of thermal conductance across boundaries will reduce the uncertainty in this parameter. Similarly, improvements in the technique of solar simulation will produce greater uniformity over the illuminated area; improvement in spectral match will come slowly, based on improved arc sources of radiant energy, and techniques using optical filters.

Interesting possibilities for automatic temperature regulation may come from materials development as a result of studies in physical chemistry and solid-state physics. One possibility lies in materials whose emittance increases with temperature, such as narrow-band absorbers and band-edge-shift materials. The increase may be conceived as occurring either as a result of a strong absorption (or emission) band in the infrared region or a shift in the absorption edge with temperature.

For the narrow band absorber the band width and its location are selected for the desired operating temperature range. Silicon oxide, for example, has a strong absorption band between 8 and 10 microns. It can be shown that this band produces roughly a 5 percent increase in emittance in the region from 0 to 60 C. In general, this effect is likely to be fairly small, perhaps attaining a 20 percent change in emittance under ideal conditions. As a result, it may be useful only as a supplementary technique.

The alternate method of a shift in the band edge is illustrated by the behavior of indium antimonide, InSb. As the temperature of InSb increases, the spectral region in which a high emittance is observed shifts reversibly to include longer wavelengths. This shift means that thermal emittance of this surface increases. Theoretically, large changes in emittance are possible and this technique would appear to merit exploration.

A second approach to automatic temperature control, which depends on a changing property, is the use of thermophototrope materials, which darken under illumination with ultraviolet radiation if they are cool enough. As they absorb solar radiation and their temperature rises, a bleaching occurs. These materials exhibit, in the laboratory, a solar absorptance which can increase and decrease reversibly. Theromphototropes, like the titanates, have been known for many years. The temperature at which they bleach has been extremely high—over 150 C. In addition, in the form they have been prepared the interesting properties are not observed under vacuum conditions. Recent work, however, has produced thermophototropes which bleach at temperatures in the 30-50 C region in approximately 30 min in the laboratory.

In addition, new groups of materials have been shown to be thermophototropic. Some of these are the oxides of zirconium, tin, cerium, lead and zinc. Techniques for enhancing the thermophototropic behavior of the titanates by the addition of impurities have also made progress. In spite of the progress, much remains to be studied. The effects of charged particles or many thermal cycles in a vacuum have yet to be established.

A reduction in temperature variation may be achieved by using materials which undergo a change of phase at an allowable temperature, thus, varying their thermal inertia. Examples of phase change which may be employed are the change from liquid to solid, and from solid to vapor. This active technique would further decrease the variation of temperature with time experienced by satellites as they enter or leave the sunlit phase of their orbits. The freezing of water can be used to prevent the cooling below 0 C for short periods (the duration would depend on the mass of water and the rate at which heat was being lost). The phase change material will impose a penalty because of its own weight and that of the confining structure, and the need to confine the material will call for reliable seals.

As weight limitations become less severe, it will become feasible to reduce gradients by the introduction of circulating fluids such as water or alcohol. By employing hollow structural members we can pump a thermally conducting fluid from the hot to the cool side of the spacecraft. These systems will require reliable pumps and seals. They will find application in satellites where large differences exist in the energy incident on different sides of the spacecraft, such as nonspinning satellites.

While potential mechanical failure in orbit makes the project manager wary of new devices, the flight tests of experimental hardware on technology satellites, such as the Applications Technology Satellite, should make the transistion to new thermal control techniques easier. ☒